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RESEARCH MEMORANDUM

THE STATIC-PRESSURE ERROR OF WING AND FUSELAGE AIRSPEED INSTALLATIONS OF THE X-1 AIRPLANES IN TRANSONIC FLIGHT

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS
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SUMMARY

Measurements were made in the transonic speed ranges of the static-pressure position error at a distance of 0.96 chord ahead of the wing tip of both the 8-percent-thick-wing and the 10-percent-thick-wing X-1 airplanes, and at a point 0.6 maximum fuselage diameter ahead of the fuselage nose of the X-1 airplanes.

For the wing-tip installation of the 8-percent-thick-wing airplane, the error increases positively, with increasing Mach number, from essentially zero at a Mach number of about 0.8 to a peak value of about 10 percent of true dynamic pressure at a Mach number of about 1.02. Then, as the wing bow wave traverses the static orifices of the airspeed head, the error drops abruptly to a negative value of 5 percent of true dynamic pressure. With further increase in Mach number, the error increases positively to a value of 10 percent of true dynamic pressure at a Mach number of about 1.32, the highest Mach number attained in this series of tests.

The error ahead of the wing tip of the 10-percent-thick-wing airplane varies with Mach number in a manner qualitatively similar to the variation ahead of the 8-percent-thick wing. The peak error is 1.20 times that of the 8-percent-thick wing at about the same Mach number. The ratio of peak errors predicted by the transonic similarity rule for wings of these thickness ratios is 1.16.

The static-pressure error ahead of the fuselage nose is about 10 percent of true dynamic pressure at a Mach number of about 0.8. The static-pressure error predicted by subsonic linearized theory for a dimensionally similar sharp-nose body of revolution is 8 percent of true dynamic pressure. With an increase in Mach number, the error increases positively to a peak value of about 22 percent of true dynamic pressure at a Mach number of about 1.07. Then, as the fuselage bow wave traverses the static orifices of the airspeed head, the error drops

abruptly to 2 percent of the true dynamic pressure and remains at this value to a Mach number of 1.17, the highest Mach number attained during this series of tests.

For the wing-tip installations (within the range $M = 0.81$ to $M = 1.00$) and for the fuselage-nose installation (within the range $M = 0.86$ to $M = 0.96$) no sensible variation of the static-pressure error occurs with airplane lift coefficient to values of airplane lift coefficient of 0.5.

The static-pressure errors are essentially free from error due to the airspeed head itself and are due entirely to the pressure field of the airplane.

At supersonic Mach numbers, a theoretical total-head correction can be applied, with sufficient accuracy, to measurements made from the nose-boom installation. In the supersonic Mach number range, to the highest Mach number attained in this series of tests ($M = 1.32$), a theoretical total-head correction can not be applied to measurements made from the wing-tip installation; the wing-tip installation lies behind a fuselage bow wave of unknown obliquity.

INTRODUCTION

An important phase in the flight testing of any aircraft is the determination of the errors involved in the measurement of the free-stream total and static pressures, as a knowledge of these pressures forms the basis for the calculation of airspeed, Mach number, and pressure altitude.

A well-designed airspeed head, such as is exemplified by current high-speed types, will not of itself cause any sensible error in either the total or the static pressures. The important errors that may occur, other than obvious misalinement of the airspeed head, arise from interference of one kind or another from the airplane structure (position error).

The total pressure may be determined accurately at any point in the flow field lying outside the boundary layer or wake of the airplane structure provided that no strong shocks of unknown obliquity lie ahead of the total-pressure port. If the obliquity of the shocks be known, the error in total pressure can easily be calculated from the shock-wave equations.

The measurement of the free-stream static pressure poses a more difficult problem, since at any point near the airplane the static pressure will, in general, be different from that of the free stream.

During the initial stages of the flight tests of the X-1 airplanes, measurements were made at Mach numbers between about 0.8 and 1.3 to determine the static-pressure position error at two locations: at a distance of 0.96 chord ahead of the left wing tip and at a distance of 0.6 maximum fuselage diameters ahead of the fuselage nose.

Past experience with numerous subsonic airplanes had shown that at these locations the position error was relatively small and reasonably constant. The measurements made on the X-1 airplanes have shown that airspeed installations designed from subsonic criterions may, at transonic and supersonic speeds, be subject to very large position errors which may vary considerably with Mach number. The results of these measurements, the first to become available at transonic and supersonic speeds, are presented herein with a view toward providing an appreciation and a qualitative understanding of the problems involved in airspeed measurement at high Mach numbers.

The data presented herein give the errors in the measurement of free-stream static pressure. Errors in the measurement of total-head pressures, within the range of Mach numbers reported, were not measured.

SYMBOLS

p'	recorded static pressure
p	true static pressure
Δp	error in measurement of static pressure $(p' - p)$
M'	Mach number corrected for static-pressure error
C_N	airplane normal-force coefficient $\left(\frac{nW}{q'S}\right)$
n	airplane normal acceleration
W	airplane weight
q'	dynamic pressure $\left(\frac{\gamma p M'^2}{2}\right)$
S	airplane wing area

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c	wing-tip chord
D	maximum fuselage diameter
d	pitot-static tube diameter

INSTRUMENTATION

Airplane instrumentation.- The pitot-static heads used in the flight tests are shown in figure 1. The NACA high-speed static orifices are spaced equally about the entire circumference of the head, and the static orifices of the Kollsman high-speed head are located on the top and bottom of the head. The drain holes in the impact-pressure chambers of both heads were sealed during these tests.

Pertinent locating dimensions of the pitot-static head for the wing- and fuselage-boom installations on the airplanes are shown in figure 2.

Measurements of static and impact pressures were recorded on a standard NACA airspeed recorder. Airplane normal accelerations were obtained from a recording accelerometer located at the airplane center of gravity. The internal recording instruments were synchronized by an NACA chronometric timer and the timing sequence, together with pressure and acceleration measurements, were telemetered to a ground station.

During tests with the wing-boom installation, the pitot-static head was connected to the recording instruments. For tests with the fuselage-nose-boom installation, the pitot-static head was connected to both recording and indicating instruments, so that there resulted a considerable increase in the volume, causing measurable lag effects in the system.

Ground instrumentation.- The NACA tracking equipment used at Muroc for airspeed calibration is an SCR 584 radar set modified for longer range. The equipment incorporates an M-2 optical tracking unit to permit remote control of the unit to eliminate the inherent hunt of the radar when tracking is done automatically. A 16-millimeter camera records radar slant range, elevation, and azimuth that are used to obtain a time history of the airplane geometric altitude throughout the flight. These data are synchronized with the airplane internal instruments through the telemeter station.

METHODS

The radar tracking method as described in reference 1 was used in calibrating the airspeed system of the X-1 airplanes. A pressure survey relating geometrical altitude and static pressure was first obtained by tracking, with radar, a B-29 airplane which towed an NACA standard trailing static-pressure bomb. The X-1 airplane was then tracked by radar through the same geometric-altitude range in a level-flight run to a high subsonic Mach number and in constant Mach number turns to determine the effects of normal-force coefficient on the position error of the airspeed installation as distinct from the Mach number effects. The data obtained thus enable an airspeed calibration to be established to a high subsonic Mach number. In succeeding flights, a pressure survey relating geometric altitude and static pressure was obtained by tracking the X-1 airplane with radar at a high subsonic Mach number for which the position error was known. The X-1 airplane was then tracked by radar through the same geometric-altitude range at higher Mach numbers. This method has been utilized in extending the airspeed calibration of the X-1 airplanes to the highest flight Mach numbers attained.

ACCURACIES

The accuracy with which an airspeed calibration can be obtained depends on the accuracy of the ground radar tracking unit in establishing geometric altitudes and on the accuracies of the internal recording instruments in measuring static and dynamic pressures. For the tests reported herein, the accuracies that can be expected from these sources are as follows:

Radar.- The geometric altitude is determined from radar elevation angle and radar slant range. With the modified SCR 584 unit in use at Muroc, several cumulative random errors may exist which can result in a maximum error in the measurement of elevation angle of ±5 mils, and a maximum error in the measurement of radar slant range of ±90 feet. It is therefore possible that a maximum error of ±600 feet may exist for any one geometric-altitude point obtained from a flight run at 20,000 to 40,000 feet at the maximum practical optical tracking range of 40,000 yards. By evaluating these data from a faired time history of computed geometric altitudes, the accuracy of determining the geometric altitudes can be increased; the accuracy of geometric-altitude measurements for the data presented herein are on the order of ±75 feet. As geometric-altitude measurements are required for both the pressure survey and the speed run, a cumulative geometric-altitude error of ±150 feet is possible in the determination of the static-pressure error. This corresponds to a maximum probable error in the determination of static pressure of about ±0.040 inch of mercury at 40,000 feet to about ±0.075 inch of mercury at 25,000 feet.

Internal recording instruments.– The internal recording airspeed instruments consist of an NACA airspeed cell with a range of 0 to 140 inches of water, and an NACA altimeter cell with a range equivalent to 0 to 50,000 feet of pressure altitude. The accuracy of these cells is ±0.25 percent of full-scale deflection. This is equivalent to ±0.07 inch of mercury in the measurement of pressure altitude and ±0.025 inch of mercury in the measurement of airspeed.

Lag.– Pressure changes resulting from the movement of fuselage and wing-bow-wave formations past the static orifices of the pitot-static head are comparable to an instantaneous change in static pressure on the airspeed system. From measurements of these changes, the method for determining the lag constant for an instantaneous change in applied pressure, as given in reference 2, has been used to determine the effects of lag on the pressure systems of the X-1 installations at altitude. It was found that, within the test altitudes flown, and for the rates of ascent and descent attained, the effects of lag upon the wing-boom installations are within the accuracy of measurement of these instruments. As both indicating and recording instruments were connected to the fuselage-boom airspeed head, there was a considerable increase in the volume of the system causing measurable lag effects. A time lag of 0.32 second was possible in the measurement of static pressures. This is equivalent to a lag of about 0.06 inch of mercury at an altitude of 33,000 feet.

As a result of the foregoing cumulative errors, the data presented herein for the wing-boom installation are accurate to the order of ±2 percent of the true dynamic pressure, and the data presented herein for the fuselage-boom installation are accurate to the order of ±3 percent of the true dynamic pressure.

RESULTS AND DISCUSSION

Accuracy of Pitot-Static Heads

The static-pressure error at 0.96 chord ahead of the left wing tip of the 10-percent-thick-wing X-1 airplane, as measured by the Kollsman high-speed and the NACA pitot-static heads, is presented in figures 3 and 4, respectively, as the variation of the pressure coefficient $\Delta p/q'$ with Mach number. The variation of the static-pressure error with Mach number measured by means of the NACA pitot head is similar to that measured by means of the Kollsman head. The absolute magnitude of the error, within experimental accuracies, can be considered the same for both heads. Data from reference 3 indicate that neither appreciable errors nor large discontinuities in static-pressure measurements were encountered in the transonic speed range in tests of an NACA high-speed pitot-static head mounted on the fuselage-nose boom of a freely falling body. Hence, it may be concluded that either head is essentially free from error of itself and that the static-pressure error is due entirely to the pressure field of the airplane.

Static-Pressure Error of Wing-Tip Installation

Variation with Mach number for 10-percent-thick wing.- The data in figures 3 and 4 indicate that at a Mach number of about 0.8, the static-pressure error is essentially zero. With an increase in Mach number, the static-pressure error increases to a peak of 13 percent of true dynamic pressure at a Mach number of about 1.02. It then drops abruptly to about 5 percent of true dynamic pressure, and again increases positively with further increase in Mach number.

The static-pressure error is a point measurement of the pressure field about the airplane and the changes in the static-pressure error reflect the changes in the pressure field at a particular point. At the point ahead of the wing tip the pressure at the static orifices is influenced positively by the wing and negatively by the fuselage. With an increase in subsonic Mach number both effects increase in magnitude, with the wing effect predominant so that the net effect is a positive increase in the static-pressure error, as is indicated.

At a Mach number of 1.0, the positive-pressure region ahead of the wing is terminated by shock in the wing bow wave. As the Mach number is further increased, the bow wave approaches the wing leading edge. The abrupt drop in the static-pressure error at about $M = 1.02$ marks the Mach number at which the wing bow wave traverses the static orifices of the airspeed head. As may be noted, the data have not been faired in the Mach number range where abrupt pressure changes occur, since the effect of angle of attack and yaw angle on the bow-wave movements has not been determined. At this Mach number and at all higher Mach numbers, the head is isolated from the field of the wing and is influenced only by the effects of the fuselage which, as is indicated in figures 3 and 4, induces a negative-static-pressure error of 5 percent of true dynamic pressure at a Mach number of 1.02.

At supersonic speeds, the pressure at a point on the fuselage is felt outwardly approximately along a Mach line which becomes increasingly sweptback as the Mach number is increased. Thus, with an increase in supersonic Mach number, the static pressure at the wing-tip airspeed head is influenced more by the positive-pressure region about the nose of the fuselage and less by the negative pressures about the more rearward portions of the fuselage. This is indicated by figures 3 and 4 where the static-pressure error increases positively with increasing Mach number, following passage of the wing bow wave.

Variation with Mach number for 8-percent-thick wing.- The static-pressure error at 0.96 chord ahead of the left wing tip of the 8-percent-thick-wing airplane is shown in figure 5 as the variation of the pressure coefficient $\Delta p/q'$ with Mach number. The variation of the static-pressure error ahead of the 8-percent-thick wing is similar to that ahead of the 10-percent-thick wing, but is slightly less in magnitude. Immediately after passage of the wing bow wave, a negative-static-pressure error of about 5 percent of the true dynamic pressure is indicated. This same negative error, due to the effect of the fuselage pressure field on the wing-tip airspeed head, will necessarily exist immediately prior to passage of the wing bow wave. Allowing for this fuselage effect, the peak static-pressure error, that which exists just prior to passage of the wing bow wave, is 15 percent of true dynamic pressure. The peak static-pressure error of the 10-percent-thick-wing installation, allowing for the same fuselage effect, is 18 percent of the true dynamic pressure. These errors are in the ratio of 1.20. It may be shown by means of the transonic similarity rule that, for airfoils with the same thickness distribution, the peak static-pressure error varies directly as the two-thirds power of the airfoil thickness ratio. The theoretically predicted ratio of peak static-pressure error for the two airfoils is 1.16 as compared to the measured ratio of 1.20.

Measurements of the static-pressure error were carried to a higher Mach number ($M = 1.32$) in the case of the 8-percent-thick-wing installation than in the case of the 10-percent-thick-wing installation, with the result that in figure 5 the effects of fuselage interference at supersonic Mach numbers can be seen more clearly. The static-pressure error reached a positive value of about 10 percent of true dynamic pressure at a Mach number of 1.32. It is to be expected that at some Mach number greater than 1.32, the fuselage bow wave will cross the static orifices of the wing-tip airspeed head. At this and all higher Mach numbers, the wing-tip airspeed head will be completely isolated from the field of the airplane, and should indicate exact free-stream static pressure.

Variation with airplane lift coefficient.- In order to show the effect of airplane lift coefficient on the static-pressure error of the wing-tip airspeed installation, the data collected on the 10-percent-thick-wing X-1 airplane have been plotted in figure 6 as the variation of the pressure coefficient $\Delta p/q'$ with C_{N_A} both for Mach numbers between 0.81 and 0.85 and for Mach numbers between 0.98 and 1.00. It is indicated in figure 6 that the static-pressure error is essentially independent of airplane lift coefficient for values up to 0.5.

Static-Pressure Error of the Fuselage-Nose

Installation

<u>Variation with Mach number</u>.- In the case of the 8-percent-thick-wing airplane, a measurement of the static-pressure error was also made at a distance of 0.6 maximum fuselage diameter ahead of the fuselage nose. These data are presented in figure 5 as the variation of the pressure coefficient $\Delta p/q'$ with Mach number. At a Mach number of 0.8, the static-pressure error was about 10 percent of true dynamic pressure. The static-pressure error as predicted by subsonic linearized theory for a point 0.6 maximum fuselage diameter ahead of a dimensionally similar sharp-nose body of revolution is 8 percent of the true dynamic pressure. With an increase in Mach number, the static-pressure error increases positively to a peak value of 22 percent of true dynamic pressure at a Mach number of about 1.07, then drops abruptly to 2 percent of true dynamic pressure and remains constant at this value to the highest Mach number attained during these measurements ($M = 1.18$).

The static-pressure error of the fuselage-nose installation arises principally from fuselage interference; the wing interference is ineffective at so great a distance ahead of the leading edge. The increase in the position error between $M = 0.9$ and $M = 1.07$ reflects the growth of the positive-pressure field ahead of the fuselage which takes place at high subsonic Mach numbers.

At a Mach number of 1.0, the positive-pressure region ahead of the fuselage is terminated by shock in the fuselage bow wave. As Mach number is further increased, the bow wave approaches the fuselage. The abrupt decrease in position error at $M = 1.07$ marks the Mach number at which the fuselage bow wave traverses the static orifices of the airspeed head. At this and all higher Mach numbers the airspeed head, since it is operating ahead of the fuselage bow wave, is completely isolated from the pressure field of the airplane and should indicate, essentially, free-stream static pressure. The cause of the indicated static-pressure error of 2 percent of true dynamic pressure after passage of the fuselage bow wave is not known.

The static-pressure error measured at the fuselage-nose position was about 10 percent of true dynamic pressure higher than that measured at the wing-tip position. This does not indicate, however, that a point ahead of the fuselage nose is a poor choice for the location of an airspeed head. The large peak error of 22 percent of true dynamic pressure that was obtained was due to the proximity of the airspeed head to the rather blunt fuselage nose. Tests made at the Langley Laboratory by the wing-flow method (reference 4), after the measurements reported herein, have shown that the peak static-pressure error of the fuselage-nose installation on the X-1 airplane may be reduced to about

5 percent of true dynamic pressure by locating the airspeed head at a distance of 1.5 maximum fuselage diameters ahead of the fuselage nose.

Variation with airplane lift coefficient.- The effect of airplane lift coefficient on the static-pressure error of the fuselage-nose airspeed installation is shown in figure 7 as the variation of the pressure coefficient $\Delta p/q'$ with C_{N_A} both for Mach numbers between 0.86 and 0.91 and for Mach numbers between 0.91 and 0.96. It is indicated, in figure 7, that within the accuracy of measurement, the static-pressure error is essentially independent of airplane lift coefficient for values up to 0.5.

Error in Total Pressure

No attempt was made to evaluate the total-pressure errors of the airspeed installations within the Mach number range investigated.

No sensible error in total pressure should be expected for either installation at any Mach number below 1. At supersonic Mach numbers, the standing bow wave which forms ahead of the pitot-static head causes a loss in total head which does not permit the total-head tube to measure the actual free-stream stagnation pressure. Data presented in reference 5 indicate that the actual loss in total head due to the formation of the standing detached bow wave corresponds very closely to the loss predicted by theory. It is possible then to apply, with sufficient accuracy, a theoretical total-head correction to measurements made from the nose-boom installation for supersonic Mach numbers. For the data presented herein, the total-head loss for the fuselage-boom installation is negligible.

Exact calculation of the error in total pressure for the wing-tip installation is more difficult, however, since below the Mach number at which the fuselage bow wave crosses the wing-tip airspeed head, this head will be operating behind a shock wave of unknown obliquity. Accordingly, the wing-tip airspeed data presented herein have not been corrected for total-head loss.

CONCLUSIONS

Measurements made of the static-pressure error at a distance of 0.96 chord ahead of the wing tip of both the 8-percent-thick-wing and the 10-percent-thick-wing X-1 airplanes indicate that:

1. The static-pressure error ahead of both the 8-percent-thick wing and the 10-percent-thick wing is essentially free from error due to the airspeed head itself and is due entirely to the pressure field of the airplane.

2. The static-pressure error ahead of the 8-percent-thick wing is essentially zero at a Mach number of about 0.8. With an increase in Mach number, the static-pressure error increases positively to a peak value of 10 percent of true dynamic pressure at a Mach number of about 1.02. Then, as the wing bow wave traverses the static orifices of the airspeed head, the error drops abruptly to a negative error of 5 percent of true dynamic pressure. With further increase in Mach number, the error increases positively to a value of 10 percent of true dynamic pressure at a Mach number of about 1.32, the highest Mach number attained during this series of tests.

3. The static-pressure error ahead of the 10-percent-thick wing varies with Mach number in a manner qualitatively similar to the variation ahead of the 8-percent-thick wing. The peak error is 1.20 times that for the 8-percent-thick wing at about the same Mach number. The ratio of peak errors predicted by the transonic similarity rule for wings of these thickness ratios is 1.16.

4. Within the range of Mach numbers between 0.81 and 1.00, no sensible variation of static-pressure error with airplane lift coefficient was noted at the wing-tip installations for airplane lift coefficients to 0.5.

5. In the supersonic range, to the highest Mach number attained in this series of tests ($M = 1.32$), a theoretical total-head correction cannot be applied to measurements made from the wing-tip installation; the wing-tip installation lies behind a fuselage bow wave of unknown obliquity.

Measurements made of the static-pressure error at a distance of 0.6 maximum fuselage diameter ahead of the fuselage nose of the X-1 airplanes indicate that:

1. The static-pressure error ahead of the fuselage nose is about 10 percent of true dynamic pressure at a Mach number of about 0.8. The static-pressure error predicted by subsonic linearized theory for a dimensionally similar sharp-nose body of revolution is 8 percent of true dynamic pressure. With an increase in Mach number, the error increases positively to a peak of 22 percent of true dynamic pressure at a Mach number of about 1.07. Then, as the fuselage bow wave traverses the static orifices of the airspeed head, the error drops abruptly to 2 percent of the true dynamic pressure and remains at

this value to a Mach number of about 1.17, the highest Mach number attained during this series of tests.

2. Within the range of Mach numbers between 0.86 and 0.96, no sensible variation of static-pressure error with airplane lift coefficient was noted at the fuselage-nose installation for airplane lift coefficients between 0.1 and 0.5.

3. At supersonic Mach numbers, a theoretical total-head correction can be applied, with sufficient accuracy, to measurements made from the nose-boom installation.

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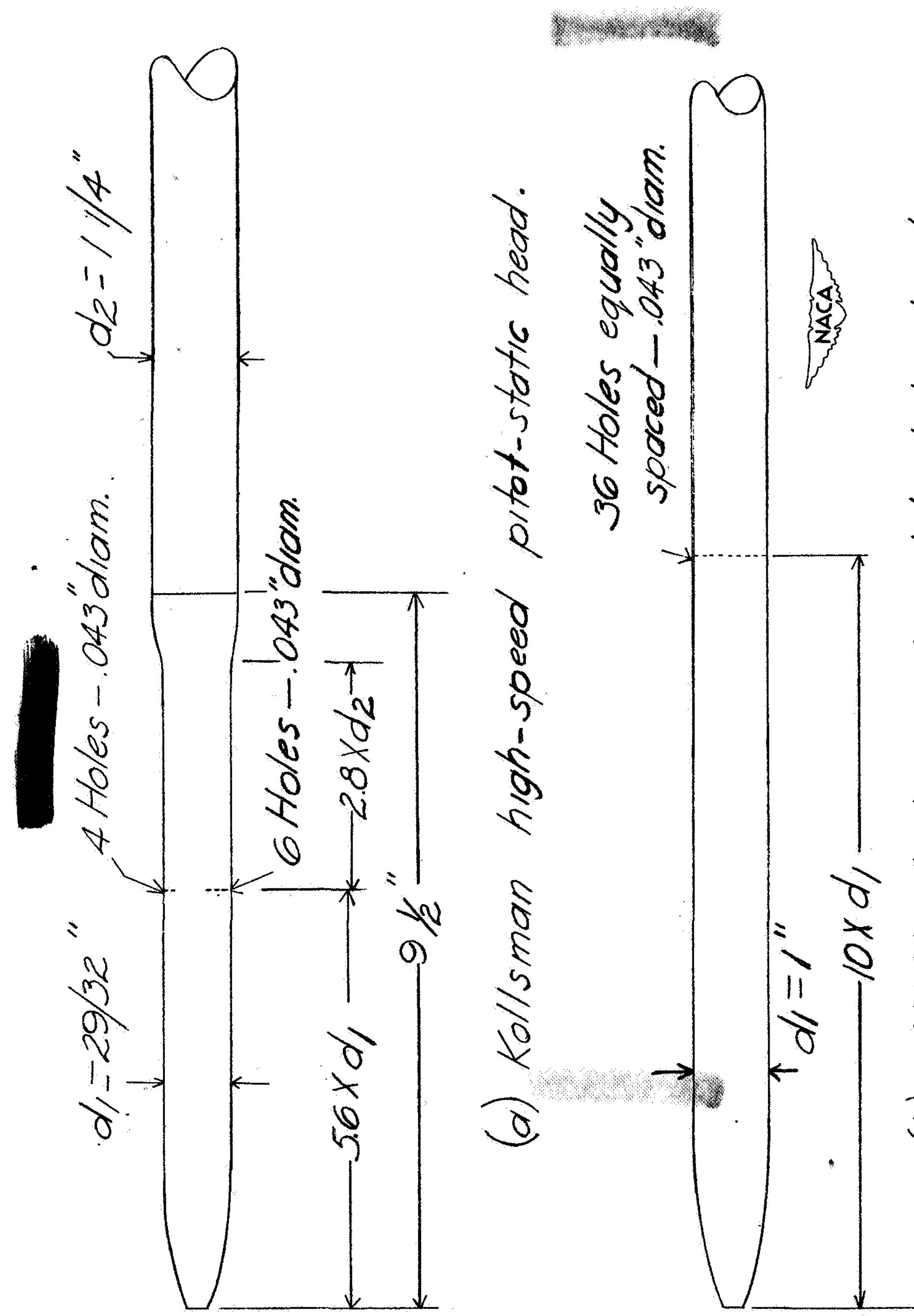

(a) Kollsman high-speed pitot-static head.

(b) NACA high-speed pitot-static head.

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Figure 1. — Kollsman and NACA high-speed pitot-static heads used on X-1 airplanes.

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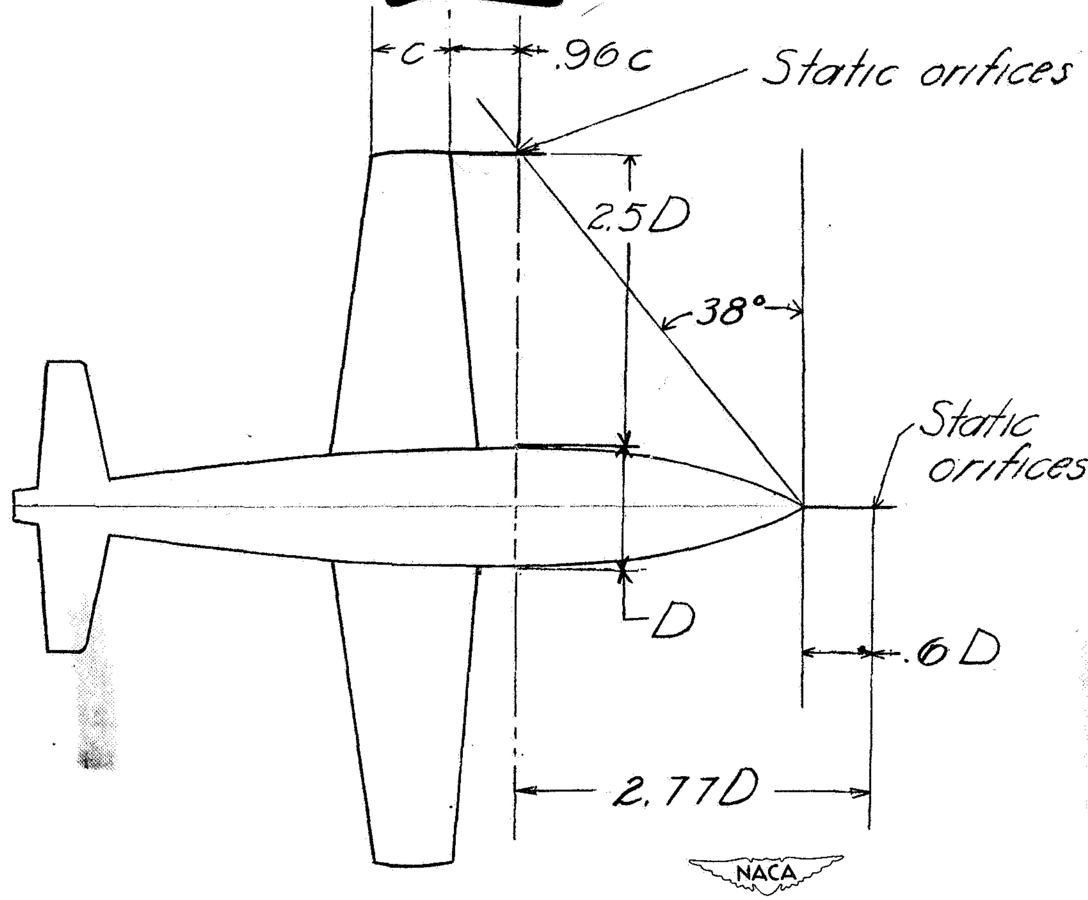

Note: not to scale

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Figure 2.— Pertinent locating dimensions of the pitot-static heads for wing and fuselage boom installations as used on the X-1 airplanes.

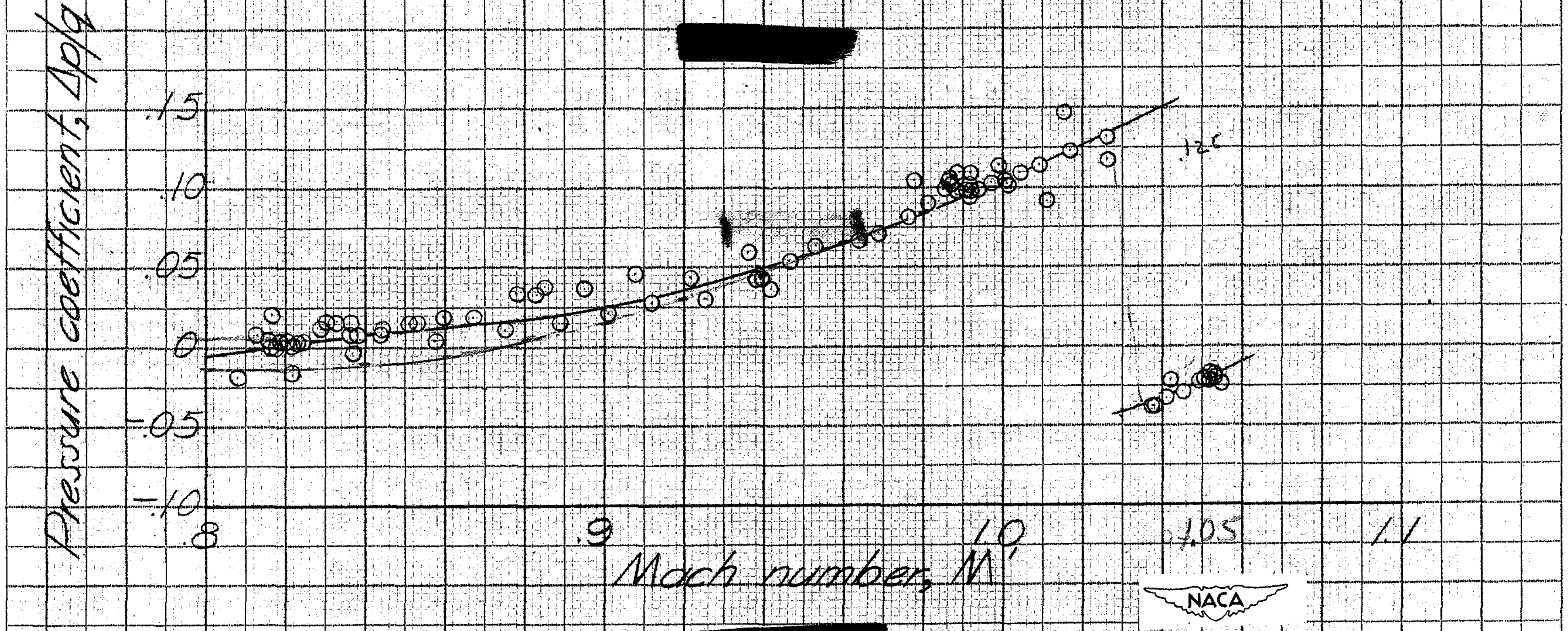

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Figure 3.— The variation of the error in the measurement of the static pressure with Mach number; Kollsman pitot-static head; wing-tip boom installation; 10-percent thick-wing X-1 airplane.

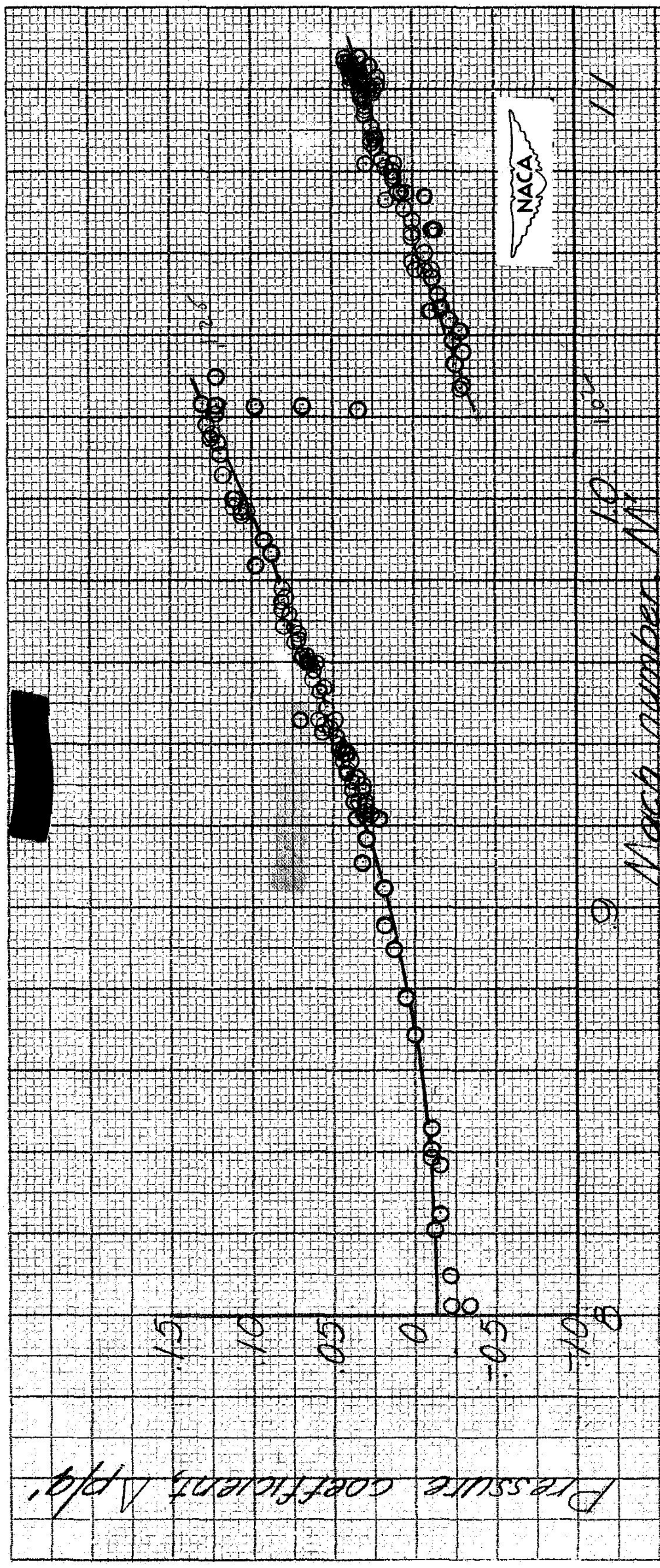

Figure 4.- The variation of the error in the measurement of static pressure with Mach number, NACA high-speed pitot-static head; wing-tip boom installation; 10-percent-thick-wing X-1 airplane.

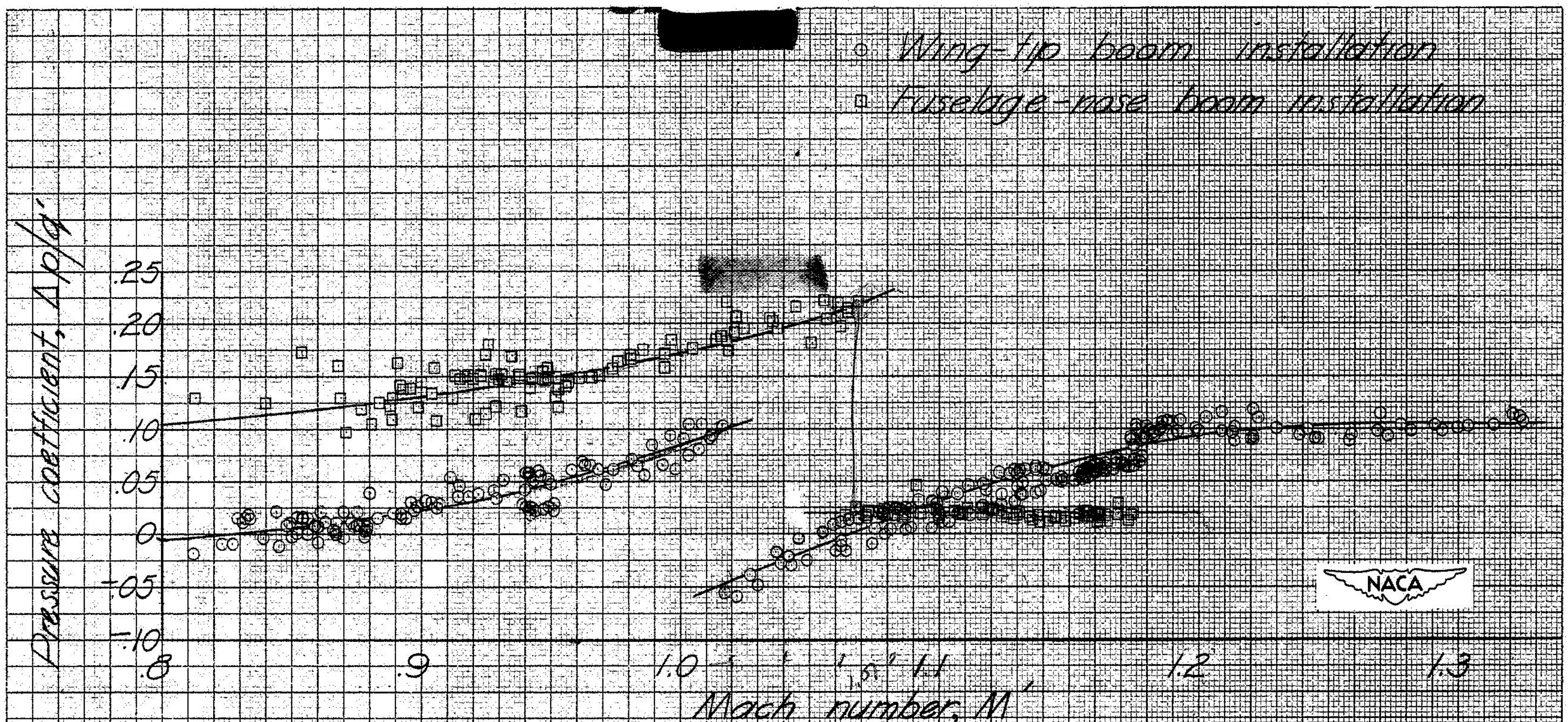

Figure 5.— The variation of the error in the measurement of static pressure with Mach number; Kollsman pitot-static head, 8-percent-thick-wing X-1 airplane.

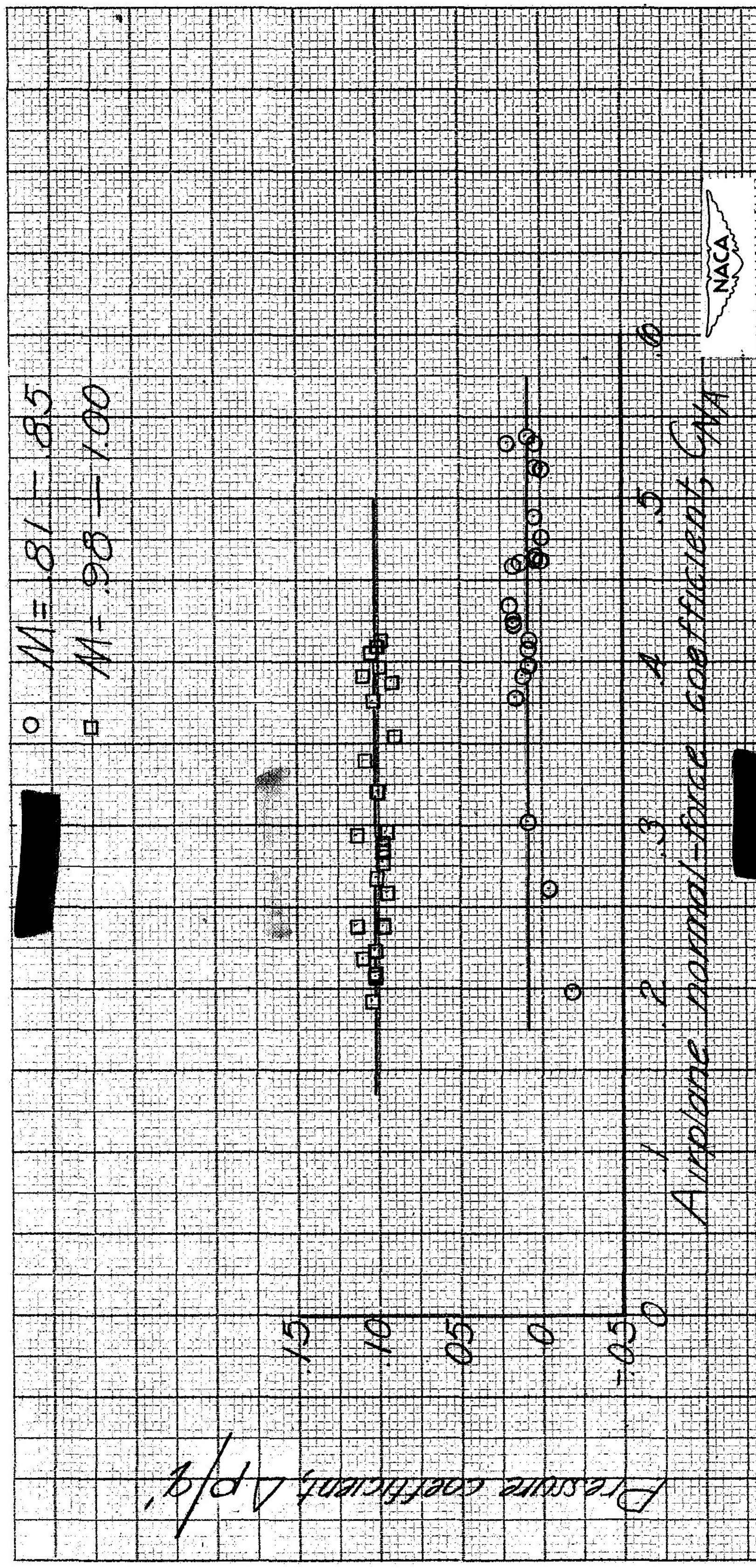

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Figure 6. The variation of the error in the measurement of static pressure with airplane normal-force coefficient; wing-tip boom installation; 10-percent-thick-wing airplane.

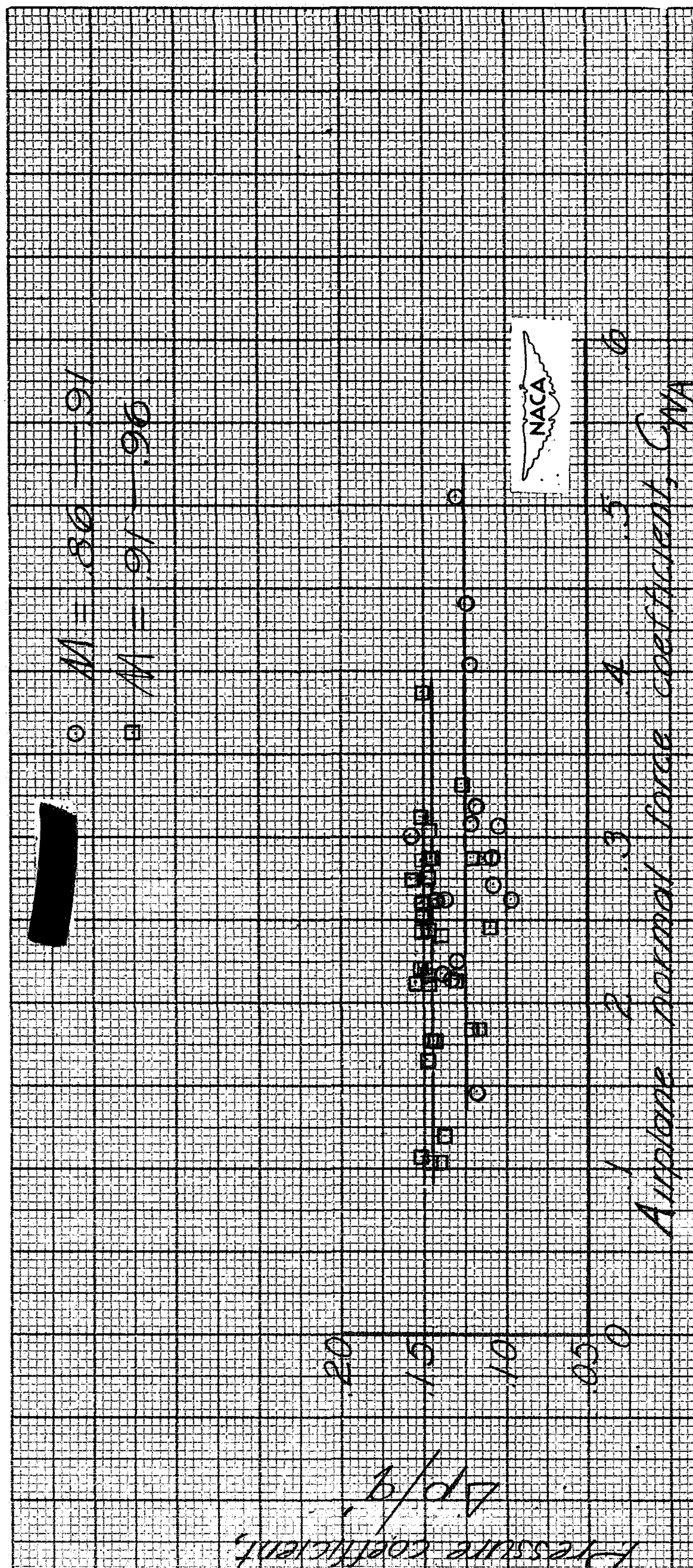

Figure 7.— The variation of the error in the measurement of static pressure with airplane normal-force coefficient; fuselage-nose boom installation; 8-percent-thick-wing airplane.